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A FEASIBILITY STUDY OF A ROCKET-POWERED HYPERSONIC WIND TUNNEL

A THESIS

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John Joseph Sullivan

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A FEASIBILITY STUDY OF A ROCKET-POWERED

HYPERSONIC WIND TUNNEL

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SUMMARY

In view of the interest in flow problems at hypersonic speeds, an inexpensive wind tunnel capable of investigation of such problems seems in demand. It is proposed that a rocket motor exhausting through a converging-diverging nozzle be used as such a tunnel. The exhaust flow would be the test medium. This study discusses the feasibility of such a tunnel, its relative cost, and its usefulness. One part of this work is the preparation of a bibliography on these and related subjects. Although all the items are not used in the discussion, they are included for future reference.

It is assumed that a flow of Mach number of five is desired and that the minimum running time is ten seconds. This time was chosen so that powder-burning rockets could be included in the discussion.

A theoretical treatment is given of the isentropic flow of a perfect gas through a nozzle, and of conditions prevailing in a rocket motor during combustion. It is concluded that under these assumptions the hypersonic flow desired is possible.

A study of available literature pertaining to rockets, propellants, and similitude points to actual limitations of such rocket use. The value of the

specific-heat ratio for rocket exhaust gases was found to be of the order of 1.2. The pressure ratio required to drive a flow of Mach number of five through the test section of such a tunnel is much higher for this value of k than for 1.4 usually used for air. To exhaust such a tunnel to the atmosphere would ideally require a rocket chamber pressure of 27,000 psia. Such high pressures have not been built into restricted burning rockets and it appears that about 6000 psia is the maximum that could be developed without obtaining detonation of the fuel. Unrestrained rockets have been fired with pressures of the order of 13,000 psia (bazooka), but the time of burning is measured in milliseconds. The mass of the fuel is one of the factors determining whether detonation occurs or not.

The addition of a diffuser on the exhaust of such a tunnel would at most permit a reduction by 13 per cent of the pressure ratio required for operation once started. It would demand a variable cross section on the throat of the diffuser which would have to be adjusted after the initial shock passed the test section. With a time of run of the order of 10 seconds it is believed not beneficial to add a diffuser.

The possibility remains of using other rockets as ejectors to reduce the exhaust pressure to about 0.25

atmospheres which coupled with a rocket pressure of about 6000 psia would theoretically give the flow desired.

It is difficult to make direct comparison of costs with conventional tunnels. The tunnel nearest in design to the proposed tunnel is one built by North American Aviation at an estimated cost of $286,000. Operating costs are not known. Its running time was twice as long; it was to achieve a Mach number near five. It, like the proposed tunnel, was intermittent in operation, but it was not known what time was necessary between runs to build the high pressure and high vacuum required. Operating cost would be a function of this time. Some comparison may be gotten by the cost of about $45 for the mass produced, powder-burning 4.5 inch rockets used by the Defense Department. The amount of fuel in such a rocket is about five pounds. A powder-burning rocket to supply the tunnel above would require about 70 pounds of powder.

Such a tunnel could be used for fundamental research but its use in plane and missile design is uncertain. The value of k for the exhaust gases is not the same as k for air. However, in air, the variation of k with temperatures encountered in flight at high altitudes is not the same as the variation of k for air in wind tunnels in use today. Relaxation time effects can be appreciable in high-speed flows. The difference in relaxation time for air and some other gases is discussed along with its effect on shock waves obtained from obstacles in the path of the flow.

The high stagnation temperature of the rocket exhaust is believed to give heat transfer similarity for model tests which is not obtained from current tests. There is the possibility of burning the metal model, shattering it by the initial shock under the high pressure, or not being able to see it due to smoke in the exhaust.

It is concluded that though its use is limited, the possibility of performing certain tests which present tunnels cannot attain and the relatively lower cost demands that the proposal be further investigated.

CHAPTER I

INTRODUCTION

It has been common engineering practice for years to use test data obtained from models and apply it to the construction of such diverse objects as bridges, canals, airplanes. This procedure has resulted in the saving of much money and labor by eliminating from design consideration undesirable features as pointed out by the model tests. Also, by model investigation, a particular shape may be shown experimentally to be the best; i.e., for a boat hull, and then the actual boat built along the same lines. (1)

Thus it is noted that many wind tunnels have been built in this country and others for the testing of airplanes and their components. Many of these have been supported by the government, some by private capital, others by universities and colleges. As well as testing the performance of possible plane designs, much work has been done in these wind tunnels to test various theories concerning the flow of fluids past objects. With the increasing interest in high speed flight as evidenced in World War II, many problems of transonic (near Mach number of one) and supersonic (Mach number greater than one) flight became important. Wind tunnels capable of test at supersonic speeds were demanded and built. The cost was high. (2)

Today a new term is used: hypersonic flight. It is flight at speeds greater than a Mach number of five, and it appears that the small educational institution is hopelessly incapable of any experimental work in this area due to the excessive cost. Need this be so?

It is suggested that a rocket be used as the source of power for an inexpensive, easy to construct wind tunnel; and further that the exhaust from this rocket be the test fluid.

The purpose of this paper is to study the feasibility of such a wind tunnel, to outline difficulties which might be encountered, and to compare costs with conventional tunnels. A complete bibliography of these and related subjects is included in the appendix.

CHAPTER II

CONVERGING-DIVERGING NOZZLE

The question of the theoretical conditions required to obtain a fluid flowing at a Mach number of five has been investigated by many authors. Following the treatment of Shapiro (3) and Sutton (4), consider a fluid flowing through a passage of varying cross section as shown in Figure 1. Label conditions at section 1 by the subscript o, and at any section 2 with no subscript. Consider an unbalance to be maintained between the two sections such that flow from left to right will occur. Define

M as V/c, the Mach number

V as the velocity of the fluid

P as the pressure of the fluid

c as the velocity of sound in the fluid

T as the absolute temperature of the fluid

A as the area of the passage

w as the mass flow rate

k as the ratio of specific heats.

Make the following assumptions:

1. The substance flowing is homogeneous in composition and homogeneous and invariable in chemical aggregation.

2. The flow is one-dimensional, steady, isentropic.
3. The velocity of flow at intake is zero, $V_o = 0$.
4. No work is extracted from the passage by shafts, pulleys, or other devices during the flow.
5. Capillary, electrical, magnetic, and nuclear effects are neglected.

By applying the first two laws of thermodynamics, the equation of continuity, definition of Mach number, and the equation of state of the fluid typical curves as shown in Figure 2 for a particular gas may be drawn.

If w, the mass flow rate is considered a constant, these curves illustrate that the cross-sectional area of the fluid passage must first decrease to a certain value, call it A_t, and then increase if supersonic flow is to be attained from the subsonic. Thus the converging-diverging nozzle is essential to the attainment of supersonic flow of a gas. If the fluid is considered a perfect gas, the following expressions may be obtained:

$$\frac{P_o}{P} = \left[\frac{K-1}{2}M^2 + 1\right]^{\frac{K}{K-1}} \qquad (1)$$

$$\frac{T_o}{T} = \frac{K-1}{2}M^2 + 1 \qquad (2)$$

These equations are plotted in Figures 3 and 4 for a k of 1.4, 1.3, and 1.2.

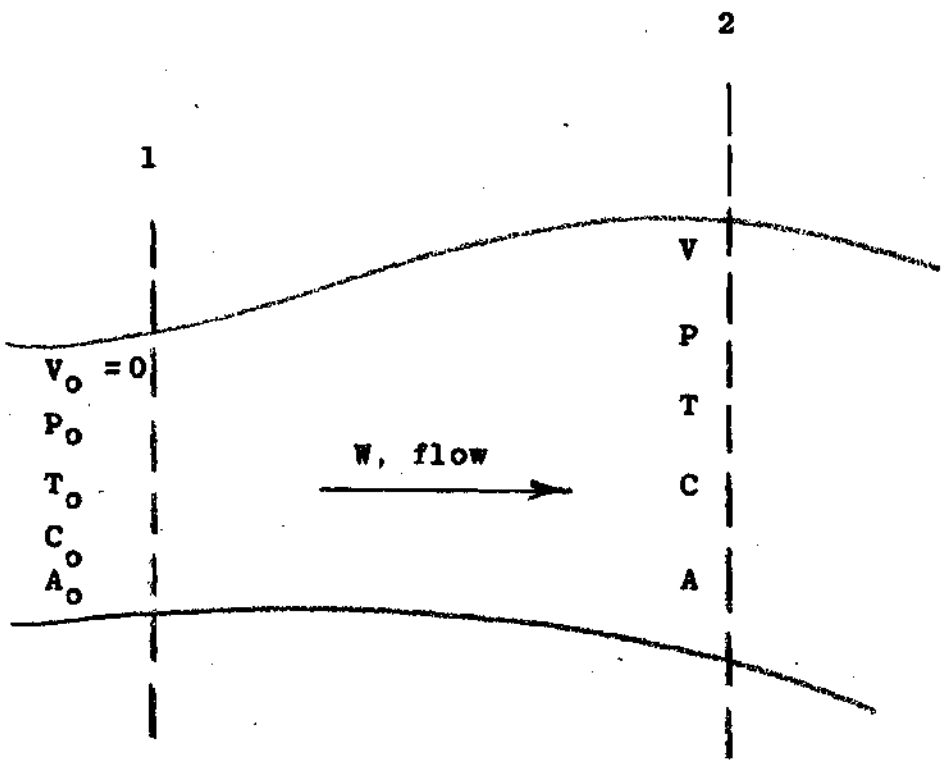

Figure 1. Flow between stagnation section and any other section

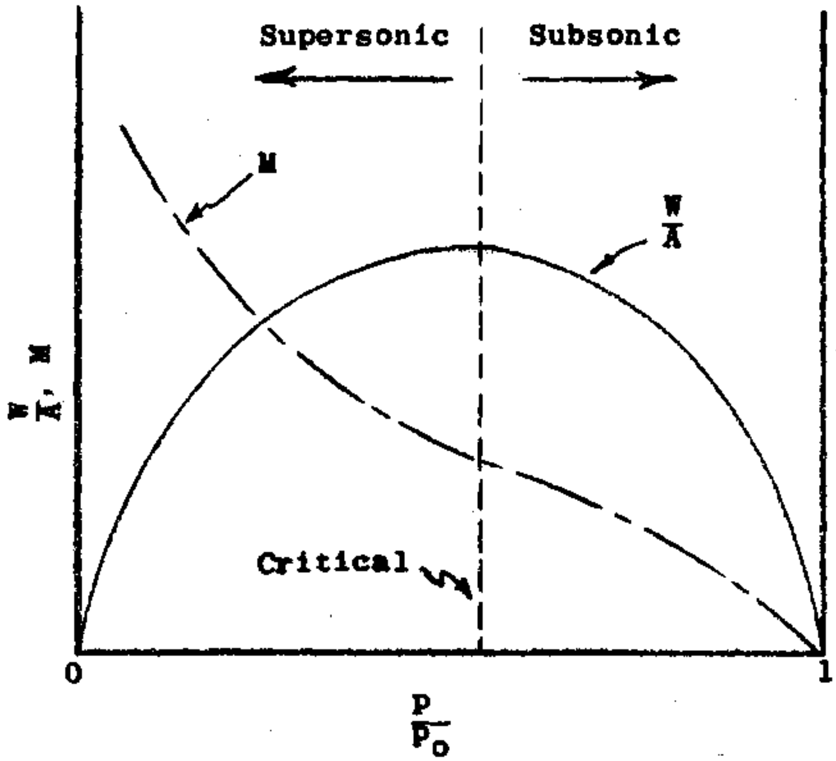

Figure 2. Typical Variation of Flow Properties in Isentropic Flow.

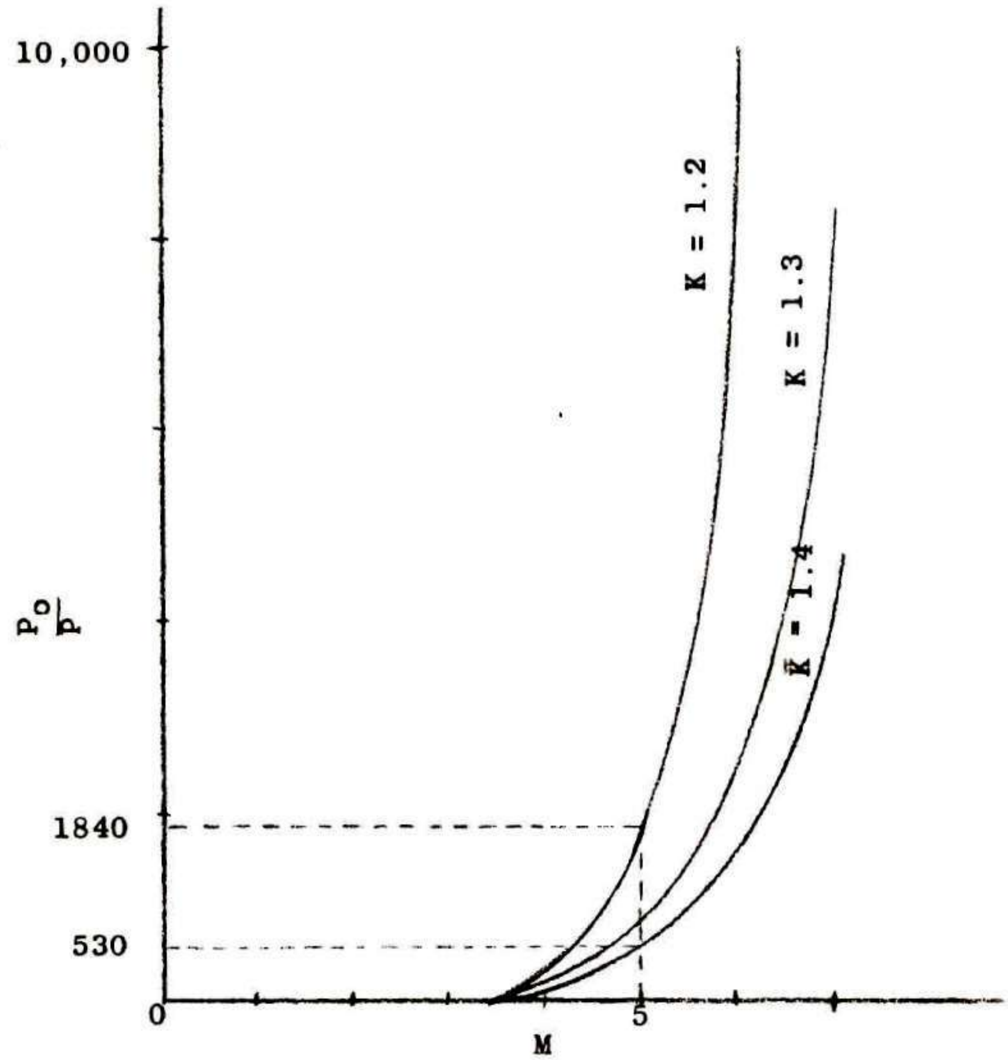

Figure 3. Pressure Ratio as Function of k and Mach Number; Perfect Gas, Isentropic Flow Assumed

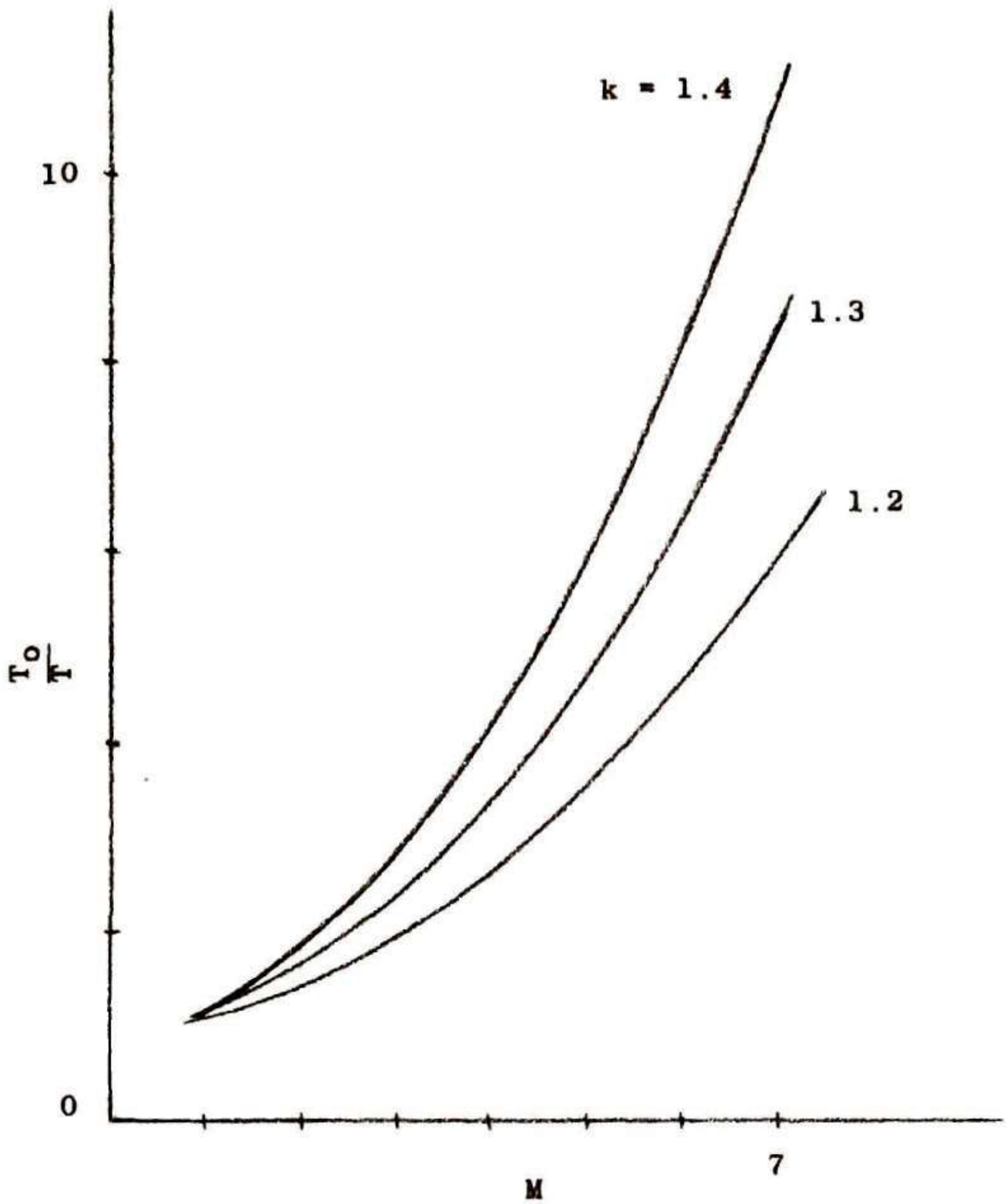

Figure 4. Temperature Ratio as a Function of Mach Number; Perfect Gas Isentropic Flow Assumed.

It is concluded that hypersonic flow may theoretically be obtained by passing fluid through a converging-diverging nozzle; and that the pressure and temperature ratios required are functions of the Mach number desired in the test section, and of k, the ratio of specific heats for the fluid.

CHAPTER III

CONVENTIONAL TUNNELS

Tunnels are in operation today which attain Mach numbers greater than five in the test section. Consider how it is done. Figure 5 shows diagramatically the elements of the simplest type of such a tunnel, the intermittent type. This tunnel is so called because the compressor and vacuum pump may be run for hours building up the high and low pressure reservoirs while the actual test takes perhaps a minute. The test consists of operating the quick-opening valve and the pressure regulator, allowing the initial shock to pass down the tunnel to the second throat, then testing in the remaining seconds available before the pressure difference becomes too small to hold the shock out of the test section. (5)

The shock that occurs when supersonic flow changes to subsonic flow is an irreversible process. The stagnation pressure drop across the shock is a measure of the irreversibility and Shapiro (6) shows that the irreversibility increases with increase in Mach number before the shock. Assuming this shock to be the only irreversibility in the wind tunnel flow, a tunnel having a shock which occurs near a Mach number of one would be more efficient than others. Hence a second throat (Figure 5) is added to

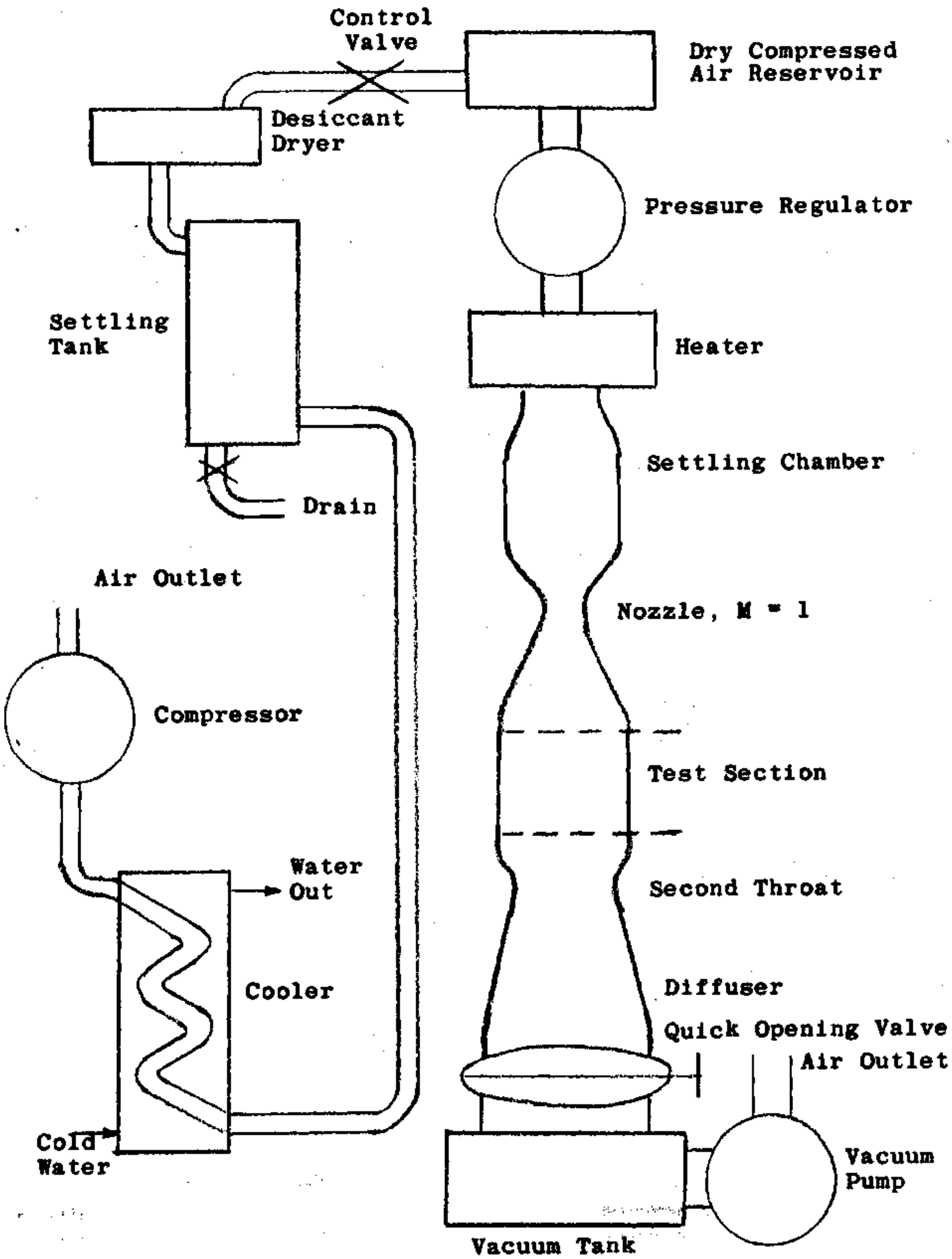

Figure 5. Conventional Intermittent Hypersonic Wind Tunnel.

compress the supersonic flow after the test section so that the Mach number is almost one before the shock. Figure 6 which shows variations in properties of a perfect gas in isentropic flow illustrates this point graphically. Holding conditions at the first throat (subscript t) and stagnation properties (subscript o) constant, a reduction of area for supersonic flow would demand an increase in pressure, temperature, and a decrease in Mach number.

When a tunnel is started a shock wave must pass through until it settles at the second throat. In its travel it encounters the test section (assumed Mach number of 5) with the maximum irreversibility. Therefore the pressure ratio required to start flow is that as discussed already for a simple converging-diverging nozzle (Fig. 3). However, once the shock reaches the second throat and the irreversibility is decreased, the overall stagnation pressure ratio should be lowered for best efficiency. (7, 8)

Another important parameter is the area of the second throat. Assuming the shock to occur adiabatically, the energy and continuity equations demand that the product of area and stagnation pressure before and after the shock must be constant. (9)

With the shock in the diverging portion of the nozzle, the first throat is passing the maximum possible flow. The second throat area must exceed the first, due to the drop in stagnation pressure across the shock. With the shock occurring in the test section (Mach number greatest)

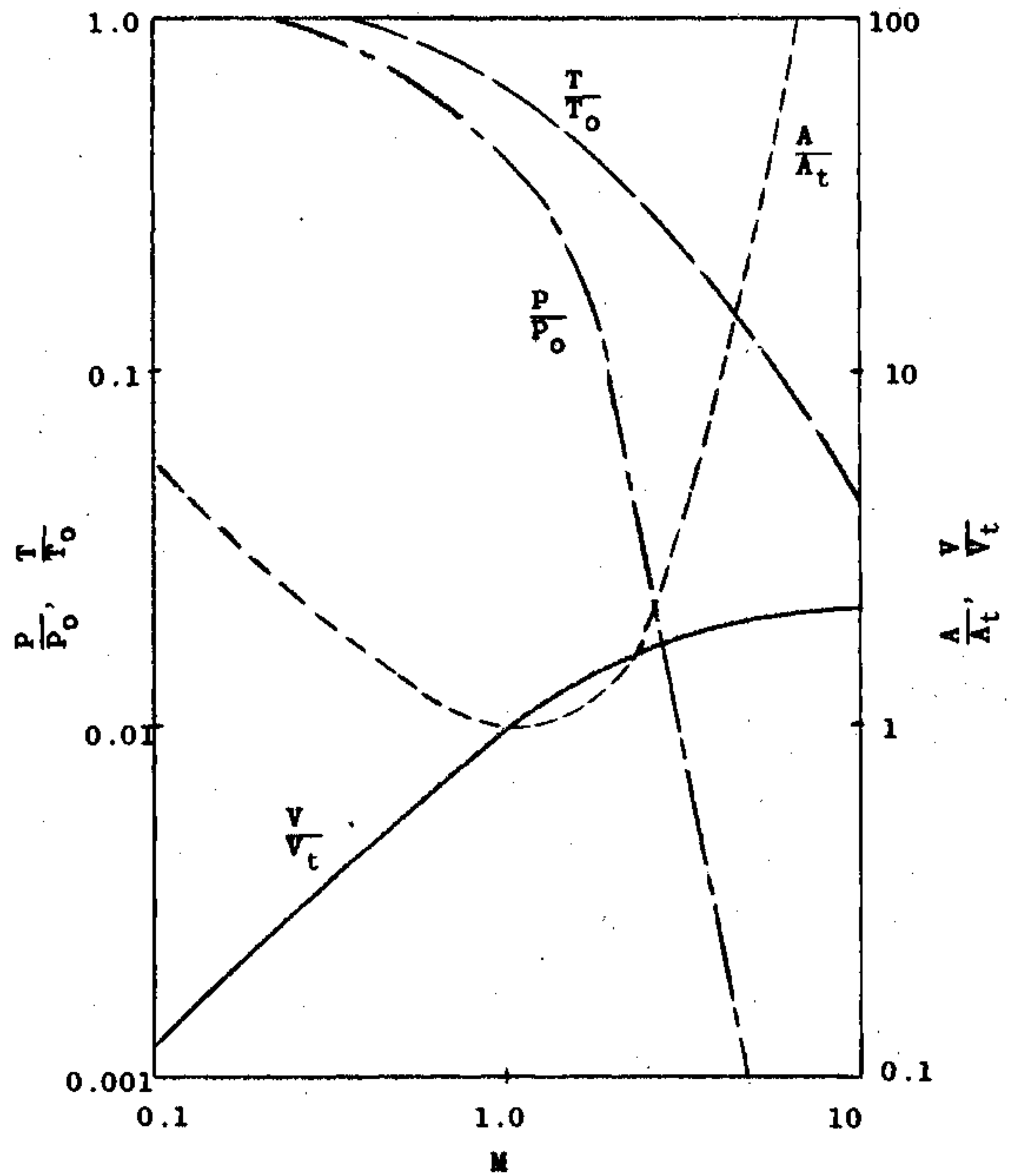

Figure 6. Working Chart for Perfect Gas, Isentropic Flow, k = 1.4

the area of the second throat will have to be still larger in order to pass the same flow, and this shock position (at the highest stagnation pressure drop) establishes the second throat minimum area for tunnel starting conditions. When the shock has reached the second throat this minimum area would have to be decreased in order to obtain a shock near a Mach number of one. This suggests that a variable cross-section of the second throat could also accomplish better efficiency, the wide throat being used for starting the flow and the smaller to maintain it.

The addition of a subsonic diffuser after the second throat acts to change some of the kinetic energy to pressure. With an ideal diffuser, isentropic flow would prevail and theoretically the stagnation pressure of the flow after the shock would be the stagnation pressure at diffuser outlet. In a continuously operating tunnel, one reason for the need for the diffuser is apparent. The tunnel must make a 360 degree loop to return the flow to its intake. Stagnation pressure losses especially in the first turn vary with the square of the velocity of flow. Should the diffuser reduce this velocity by a factor of ten, the resultant power loss would be reduced by a factor of 100. For a blowdown tunnel as proposed (Figure 7), the absence of the subsonic diffuser section would mean the fluid would exhaust to the atmosphere at a velocity slightly below sonic following the shock in the second throat. An irreversible process would follow in the

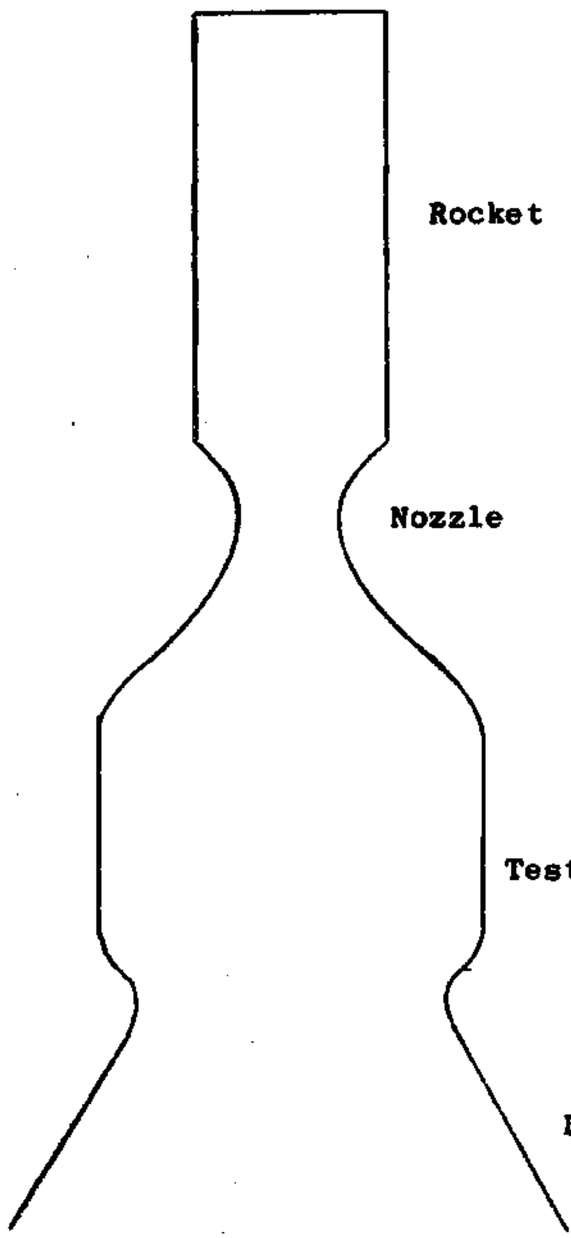

Figure 7. Proposed Tunnel

turbulence of the air as the fluid comes to zero velocity. Recalling that the stagnation pressure ratio must overcome the irreversibilities of the flow, a higher pressure ratio would be required to maintain the desired hypersonic test section velocity. A reversible diffuser would lower the pressure ratio by the pressure it recovered from the kinetic energy. (10)

Actual diffusers are not reversible. It is possible that friction of flow in the diffuser would offset the possible gain. Ferri and Bogdonoff (11) state that in general no diffuser is used in intermittent tunnels for Mach numbers below 3.5.

Neumann and Lustwerk (12, 13) have experimented with various diffuser shapes, both fixed and variable cross-sectional area. In comparing their experimental results with theoretical a maximum diffuser efficiency, η_D, of the order of 60 per cent (M=5) is plotted. It is for a variable geometry diffuser, isentropic, two-dimensional flow. These results include the second throat already discussed as an integral part of the diffuser. Using the definition of diffuser efficiency as the ratio of the isentropic enthalpy drop to the change of kinetic energy of the flow, assuming the exhaust velocity to be negligible, Shapiro (14) relates η_D to the ratio of stagnation pressures as follows:

$$D = \frac{\left[1 + \frac{k-1}{2} M_1^{\,2}\right]\left[\frac{P_o \text{ exit}}{P_o \text{ inlet}}\right]^{\frac{k-1}{k}} - 1}{\frac{k-1}{2} M_1^{\,2}} \tag{3}$$

where M_1 is the Mach number before the shock

P_o inlet is stagnation pressure at diffuser inlet

P_o exit is stagnation pressure at diffuser exit

Using this expression, taking $= 60\%$, $M_1 = 5$, $k = 1.2$, the stagnation pressure recovery $\frac{P_o \text{ exit}}{P_o \text{ inlet}}$ is about 13 per cent. This suggests that once the flow is started the pressure ratio to maintain it may at most be 13 per cent less than that shown in Figure 3.

As shown in Figure 5 the air is cooled after being compressed in order to remove the moisture. Moisture can be a problem since the test section temperature is low compared with stagnation temperature (Fig. 6) and condensation can not only fog the section but change the character of the shock waves formed on the model. (15) (16) Hence, a desiccant may be used after the cooler and settling tank to ensure that the air is dried sufficiently.

As the air flows through the pressure regulator it is throttled; hence, it is cooled. The purpose of the heater is twofold. It not only heats to ensure a constant test stagnation temperature, but also to prevent condensation of the oxygen and nitrogen of the air in the cold test section at hypersonic speeds. (17) In Figure 8(a) are shown typical curves of stagnation pressure and temperature (measured in the settling chamber) as functions of time under the assumption the pressure valve is opened wide and there is no heat addition. In (b) is shown the rate of opening the valve, and the rate of heat addition for a given mass flow rate. In (c) is shown the expected results.

Various techniques are used to obtain the necessary data. Schlieren pictures or shadowgraphs may be used. Runs of the order of 30 seconds are not unusual. (18)

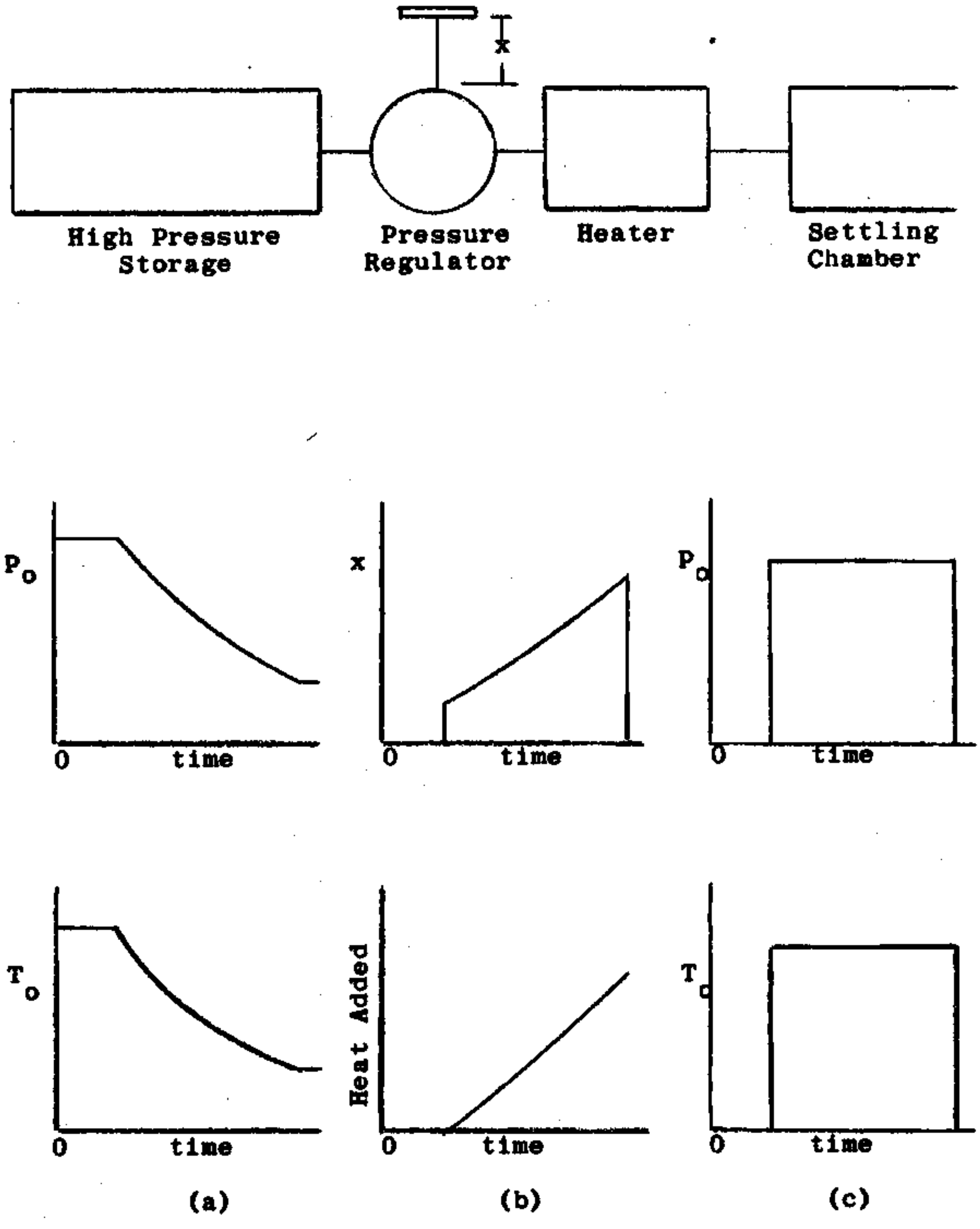

Figure 8. Desired Performance of a Pressure and Temperature Controlled System.

CHAPTER IV

ROCKETS

A rocket is essentially a combustion chamber with a nozzle on one end. It differs from a jet engine in that it does not require oxygen from the atmosphere for combustion but carries its own either mixed with the fuel (monopropellant) or as fuel and oxidizer, both of which must be fed into the chamber. The propulsive effect of a rocket is a consequence of Newton's second law which states "for every action there is an equal and opposite reaction." On combustion, the gaseous products formed tend to increase the pressure in the chamber so that they stream out the exit. This action of gas out one end causes a reaction tending to propel the rocket. Were there no nozzle on the end there still would be a driving thrust. It can be shown that the thrust is a maximum when the gas is expanded isentropicly by means of the nozzle such that the pressure of the exhaust gases equals the pressure of the surrounding atmosphere. (19)

Rocket propellants are either powder or liquid type. The theoretical action within the motor is similar in both cases. The powder propelled motor will be discussed and then the liquid type compared to it.

McClure and Kershner (20) give a theoretical development of what occurs in a rocket chamber while the propellant powder is burned. A summary of the assumptions made and conclusions drawn is listed below.

The five assumptions on pages 3 and 4 for steady, one-dimensional, isentropic flow through the nozzle are applicable. Continuing this development, assuming a perfect gas flowing, and substituting P_c, chamber pressure, for P_o gives

$$\frac{d\ (M_d)}{dt} = \left(\frac{2}{k+1}\right)^{\frac{k+1}{2(k-1)}} \frac{\sqrt{k}}{\sqrt{nRT_c}}\ A_t\ P_c \tag{4}$$

where $\frac{d\ (M_d)}{dt}$ = the time rate of mass discharge through the nozzle

A_t = area of the minimum section of the nozzle (throat)

P_c = pressure in rocket combustion chamber in psia

T_c = temperature in rocket combustion chamber in ^oF abs

n = number of moles

R = universal gas constant

Additional assumptions are:

6. Considering T_c as constant, Equation (4) may be written as

$$\frac{d\ M_d}{dt} = C_d A_t P_c \tag{5}$$

where C_d is a coefficient of discharge.

7. Assume $\frac{d\ M_b}{dt} = S \rho r$ (6)

where S = the area of the surface of the powder

ρ = the density of the solid powder

r = the velocity of travel of burning surface

$\frac{d\ M_b}{dt}$ = the time rate of mass burning

S is a function of the distance burned and can be found by geometry from the initial shape of the grains. Note in Figure 9 a single-perforated grain will have almost a constant S since burning from the inside tends to increase S while outside burning has the opposite effect. Burning of the ends of the grain decreases S and it is this effect that is named regression of the burning surface.

8. Assume r is a function only of P_c; in particular,

$$r = a + b\ P_c \tag{7}$$

where a and b are constants depending on the composition of the powder used, and the temperature of the powder before burning.

9. Assume the volume of the rocket combustion chamber available to the gases generated during the burning is

constant. Then the curves of $\frac{d\ M_b}{dt}$ and $\frac{d\ M_d}{dt}$ for actual powder propellants appear as shown in Figure 10 for a typical powder. Thus if the pressure is below a certain value, call it P_{eq} (the intersection of the two curves), the rate of discharge is less than the burning rate, and the pressure increases. Similarly, if the pressure is greater than P_{eq}, then the discharge rate exceeds the rate of burning tending to decrease the pressure in the rocket motor. P_c in effect becomes P_{eq}. By setting the burning rate equal to the discharge rate the following equation is obtained:

$$P_c = \frac{K\rho a}{C_d - K\rho b} \tag{8}$$

where $K = S/A_t$.

All of the assumptions made have a degree of validity. There are many other factors not mentioned here that affect the value of P_c but for a preliminary guide in the design of new rockets Equation (8) is used. A remarkable fact to be noticed in this equation is that all of the quantities C_d, ρ , a, b, depend only on the composition of the powder. Thus the size or structure of the rocket itself and the amount of powder used enter only in the ratio called K.

Another fact to be noticed about Equation (8) is that the denominator is the difference of two factors. In practice these are nearly equal; thus P_c is very sensitive

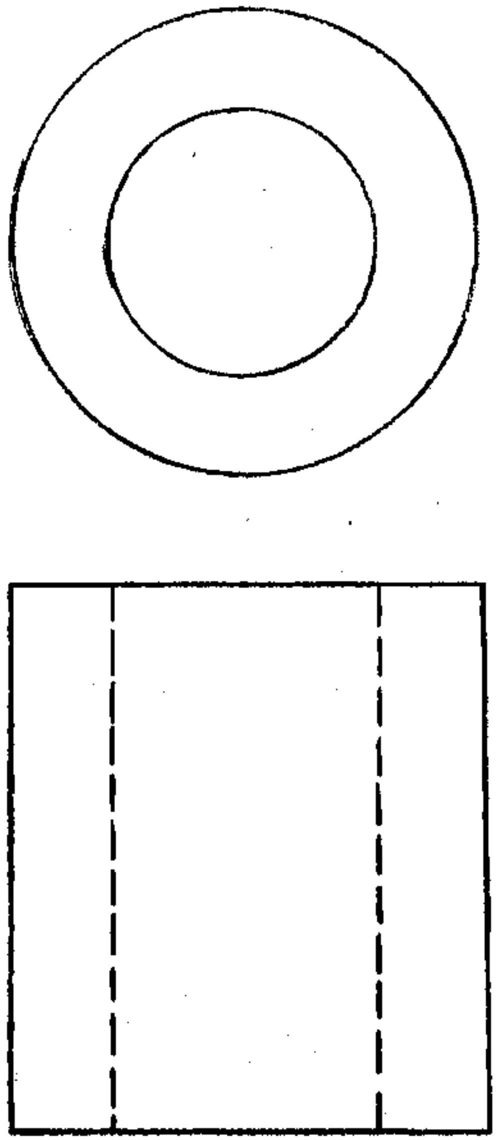

Figure 9. Single Perforate Grain

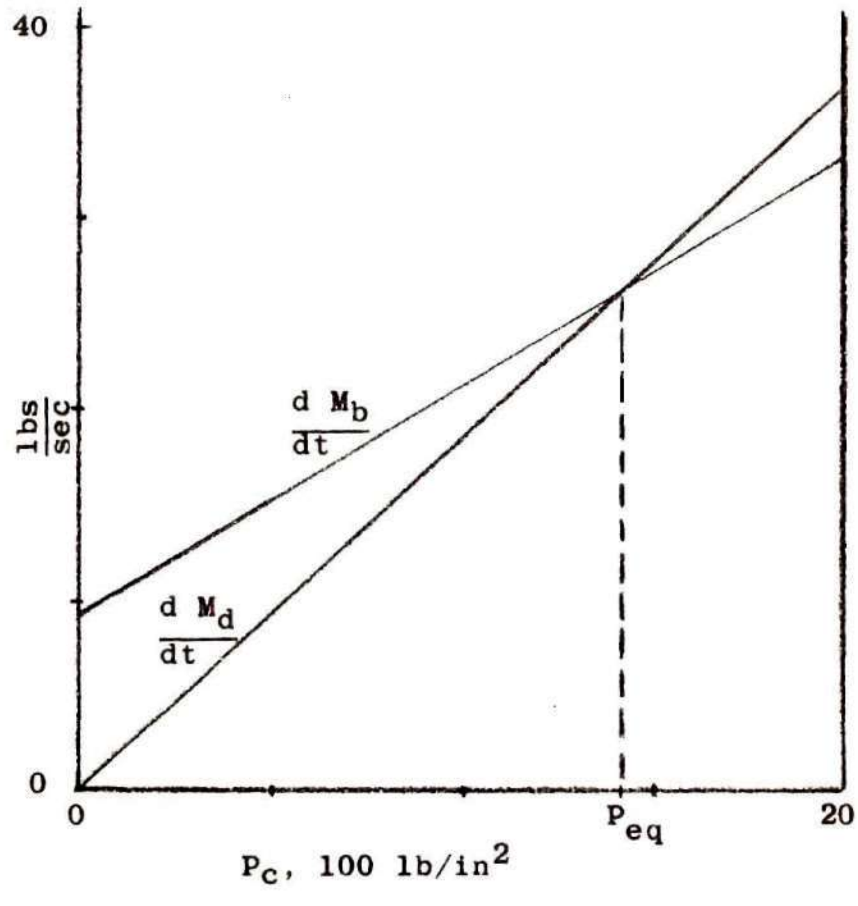

Figure 10. Comparison of Mass Burning Rate and Mass Discharge Rate as Functions of Pressure.

to changes in the burning rate and the value of K. This is not surprising since the pressure in the rocket is determined by the very small difference between the large amount of powder gas created and the almost equally large amount of gas that has escaped to the rear.

Some experimental P_c-K curves for three different ambient temperatures are shown in Figure 11. If there were no reason to prohibit the extrapolation of these curves, a large K could feasibly be selected to give any desired P_c. Figure 11 points to the acute dependency of pressure in a rocket on ambient temperature. Major difficulties were encountered in the development of military rockets that were to be just as good in Alaska as they were in the jungle. Much work has been done to decrease the sensitivity to temperature and some of the more recent propellants are greatly improved. (21, 22, 23, 24)

To apply the above discussion to liquid propellants, the time rate of mass burning would have to be replaced by the time rate of fuel injection. The rate of injection would be under the control of an operator. Then by equating the injection rate to the discharge rate the following would apply:

$$\frac{d\ M_i}{dt} = \frac{d\ M_d}{dt} = C_d\ A_t\ P_c$$

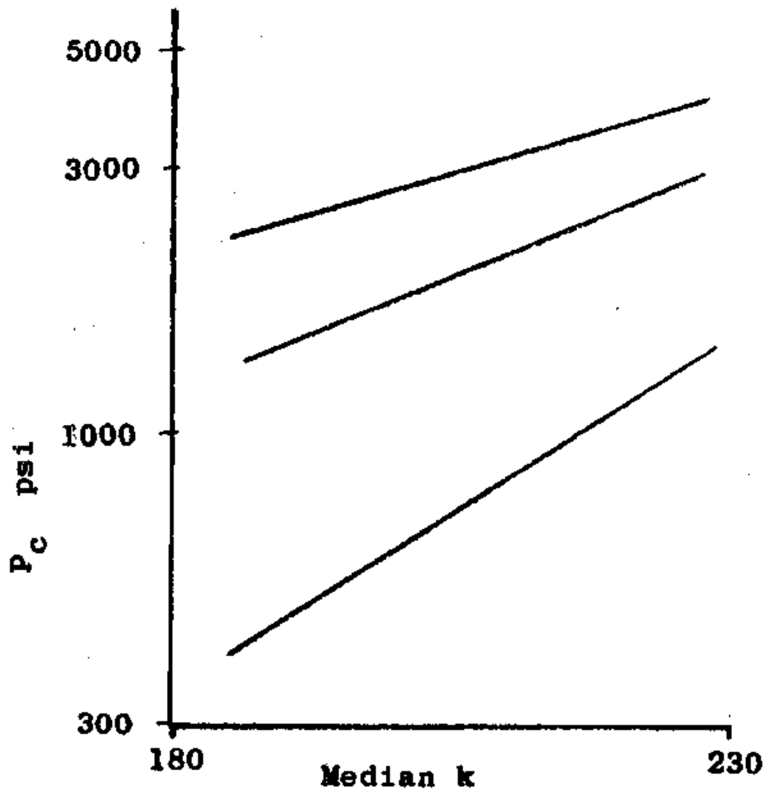

Figure 11. Experimental P-k Curve

$$P_c = \frac{\frac{d\ M_i}{dt}}{C_d\ A_t} \tag{9}$$

where $\frac{d\ M_i}{dt}$ = time rate of fuel mass injection.

Theoretically any P_c could be attained by increasing $\frac{d\ M_i}{dt}$ or by decreasing the $C_d\ A_t$ product.

Sutton (25) outlines a method of obtaining a theoretical estimate of T_c. First from the heats of formation of the reactants, corrected to bring the reactants up to a reference temperature T_i, the heat of reaction $\left[Q_r\right]_{T_i}$ at this T_i is calculated. Chemical equilibrium in the combustion chamber is assumed. Then the heat of reaction of propellants is equated to the heat capacity of the product gases:

$$\left[Q_r\right]_{T_i} = \sum n_p \int_{T_i}^{T} C_p dT \tag{10}$$

where $\left[Q_r\right]_{T_i}$ = Heat of reaction of propellant combination at T_i

n_p = moles of product gas

$\int_{T_i}^{T} C_p dT$ = enthalpy change necessary to heat one mole of each product gas from T_i to flame temperature T

C_p = molar specific heat at constant pressure for a particular product gas

T = Reaction temperature

This equation solved for T gives the combustion temperature. To satisfy equilibrium conditions when the products formed may be as many as eight chemical compounds or dissociated products, this may become a solution of 11 equations with as many unknowns. These have been solved for most of the propellants considered for rocket use. (26) The Battelle Memorial Institute currently is working on such a project for the Air Forces. A conservative average temperature is 4500°F with 5000°F not uncommon.

From a consideration of the theoretical material so far presented, it appears that as long as a specified pressure and temperature ratio can be maintained for the required 10 seconds, a test section flow of Mach number five would result. In order to maintain this pressure ratio the burning rate of propellant must exceed the discharge rate of gases formed by a calculable amount for a given time length. For powder propellants this time length is dependent on the mass of the charge in the motor. With liquids, the time length is in the hands of the operator.

It is concluded that the requirements of hypersonic flow for 10 seconds determine the surface area, charge mass, and chamber dimensions for powder propellants. For liquids the rate of fuel injection, mass to be burned, and chamber dimensions are determined. A sample calculation based on some assumed typical figures is included in the appendix. Theoretically such rocket use is feasible.

CHAPTER V

DIFFICULTIES IN DESIGN

Rockets in use today have been designed for a specific purpose. For instance, the diverging portion of the nozzle on rockets for flight is usually not greater in diameter than the main tubular portion of the motor. If the diverging portions were, it not only would cause increased drag in flight, but also would require a launching platform rather than the launching tube used with rockets such as the bazooka. In the bazooka, rocket pressures of the order of 13,000 psi are obtained but the burning time is of the order of milliseconds. This is required to make the time of flight a minimum and to ensure completeness of burning before the rocket leaves the launcher as protection for the face of the firer. (27) The weight of a rocket designed for flight is an important parameter both from total weight and weight distribution standpoints. Thus factors exist which must be considered in the design of a rocket-powered wind tunnel which would not be applicable to rockets in use today.

The available literature concerned with rockets, wind tunnels, and allied subjects has been studied by the author and design difficulties will now be considered under five headings.

<u>What is the value of k in the products of combustion?</u>

From the theoretical considerations of Chapter II it is seen that the value of k is an important parameter in the proposed rocket use. The attainment of Mach number of five in the test section under the assumption the tunnel exhausts to atmospheric pressure demands a pressure in the rocket chamber of 7800 psi for a k of 1.4; 27,000 psi for a k of 1.2.

It is known that the value of k varies with temperature. Many investigators have calculated the composition of various products of reaction for rocket propellants (liquid and powder) in use and contemplated. From these calculations theoretical values of k for exhaust gases have been published. Osborne, McClure, and Hirschfelder (28) have done much work with powder propellants and their published figures range from k of 1.13 to 1.28. For design purposes a figure of 1.2 is usually used. (29) For liquid propellants, 1.2 to 1.25 is used. (30), (31)

<u>What T_c is required to avoid condensation of fluid in the test section?</u>

As fluid flows through the converging-diverging nozzle the temperature drops as well as the pressure. This was shown for the perfect gas in isentropic flow in Fig. 6.

Should this temperature drop low enough there is the likelihood of condensation of first the moisture in the air, and secondly, of the elements of the air itself. This has been investigated by Stever and Rathbun (32) and by Willmarth and Nagamatsu (33) at the California Institute of Technology where the Navy and Army Ordnance have joined in the construction of a continuous-operating hypersonic wind tunnel. In some cases the effect is a condensation shock; in other cases the static pressure increases (Mach number decreases) as condensation begins. Investigators detected the condensation by the change in shock angle for an eleven degree wedge, or by light scattering techniques.

In general, evidence points to a minimum stagnation temperature of 800°F to avoid condensation of air in the test section for a Mach number of five. (34)

<u>What temperatures and pressures exist in rockets today?</u>

Many investigations have been made of the temperatures in rocket motors during burning. In general a range of 3500° to 5500°F may be considered to include most propellants. With hydrogen peroxide as fuel they are somewhat lower. It is concluded that temperature ratios are currently high enough to achieve hypersonic flow in the proposed tunnel with no condensation problem.

When speaking of pressures, liquid and powder propellants must be considered separately. Powder propellants are used for rockets having burning times of about 45 seconds or less. (35) It must be remembered that in powder-propelled rockets all the fuel is in the motor during burning. Thus the heat liberated by the first fuel burning is available to heat the remaining powder. Other effects become troublesome. In the development of powders for rockets, some were stored and noticed to darken in color. When these were fired, fewer ruptured motors resulted. An investigation showed that the radiation of heat from the hot gases produced in the chamber was causing high-temperatures locally about impurities in the interior of the propellant. Since darker color (aged powder) absorbed the radiation at the surface, coloring agents are frequently added to powders today. (36) Later this radiation effect was put to good use in order to obtain pressure-time curves closer to the ideal. The ideal for a rocket motor is the same as the ideal for a wind tunnel (See Fig. 8c). Actual pressure-time curves approach the ideal closely. (37) In the bazooka, rocket pressures of 13,000 psi for a time measured in milliseconds are reasonable. For longer burning rockets, the highest pressure discussed in the literature is that for a Jet Assist Take Off Unit (JATO) produced by Aerojet Corporation which burned for 12 seconds at a pressure of 1800 psi. (38)

Powder burning rockets are not only limited by their variation in performance with ambient temperature as mentioned previously, but also by the fact that the actual specific impulse obtained from them is usually less than 100 seconds. (39) Specific impulse is defined as impulse/pound of fuel burned; in units it is lb-force-seconds/lb weight of fuel. It is usually spoken of in terms of seconds. It is an important parameter in all rockets which are to be used in flight since the weight of fuel limits the range available. Specific impulses greater than 200 seconds are not uncommon with liquid fuels. Hence the long range rockets built today burn liquids. (40)

In liquid rockets the pressure commonly used is 300 psi. The reason for this is shown by Figure 12. The increase in specific impulse for a given increase in pressure is great up to 300 psi. Then the curve tends to flatten out. While there still is an increase above 300 psi the motor wall must be made stronger to accommodate the higher pressure; also the pumps to feed the fuel and oxydizer into the chamber must develop more head, all meaning added weight and the possibility of no net gain in performance. Liquid rockets have been built and experimentally tested (not flown) to operate at 700 psi. It is proposed to test run others at pressures of 2000 psi; Purdue University has a contract under Project SQUID to do the work. (41)

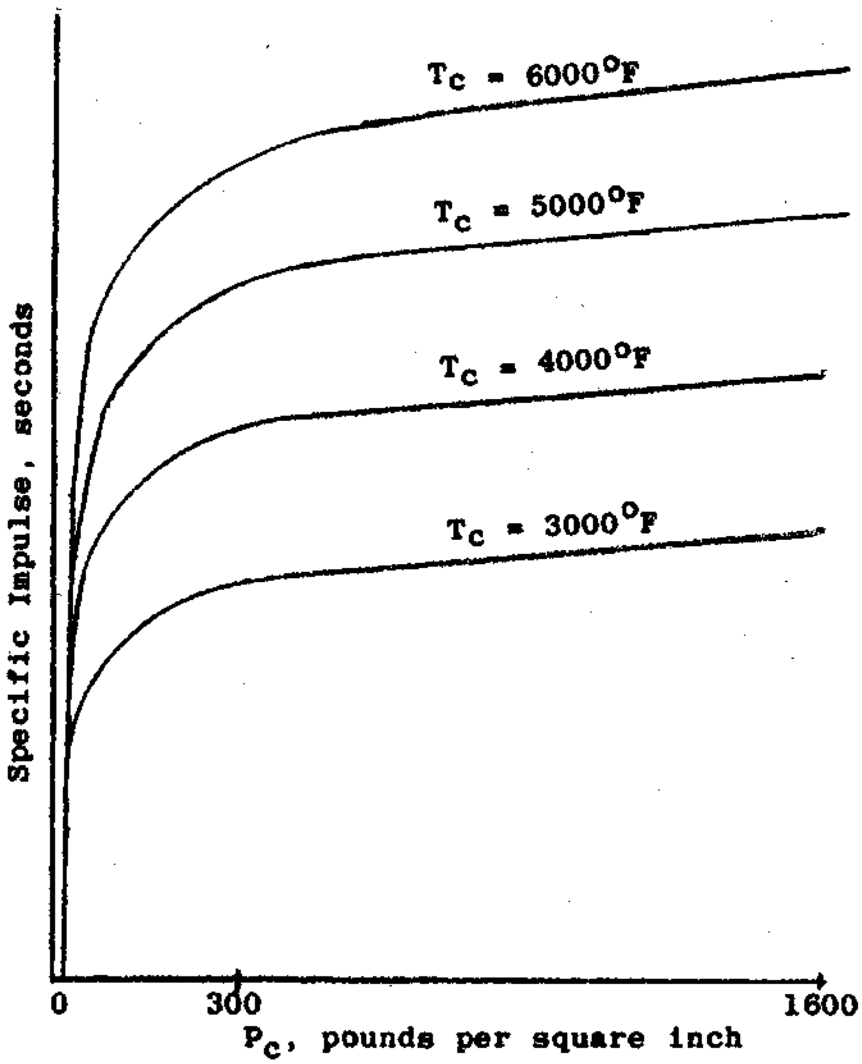

Figure 12. Effect of Combustion Chamber Pressure and Temperature on Specific Impulse.

<u>Are there limiting factors affecting maximum P_c attainable?</u>

In the theoretical treatment of combustion in rockets it was assumed that the rate of burning of powder propellant in the motor was a linear function of the pressure in the chamber. It was concluded that the pressure attained in the chamber was the result of the difference between the rates of gas generation and gas discharge.

Different propellants have different burning rates. The rate of discharge is a function of the pressure and temperature in the chamber, and the geometry of the rocket chamber, in particular, the nozzle.

Theoretically it seems feasible to pack into the rocket enough mass so that a high burning rate may prevail for the required 10 seconds. Thus, any pressure is attainable in the rocket. In rockets such as the bazooka where pressures of 13,000 psi are attained, the burning is called unrestricted. Here the entire surface of the powder is ignited simultaneously. Where a rocket is required to burn for time lengths of the order of seconds, controlled burning must be used. A common type of controlled burning is called cigarette burning whereby one end of the powder charge is ignited and the flame front travels to the other end of the rocket gradually.

Sutton (42) states that for most powder propellants the burning rate will increase so rapidly for pressures above 6000 psi that a detonation would occur rather than burning, shattering the chamber.

This leads to a discussion of what causes detonation. Kistiakowsky (43) drew upon experimental work with explosives and formulated a theory of detonation. Explosives are usually classified as primary explosives, those which detonate instantly on ignition; high explosives, those which detonate when properly initiated; and propellants, those which do not detonate. Experimental evidence points to the fact that any explosive may be deflagrated or detonated under proper conditions, and in this respect all are alike.

A theory is proposed to explain the change from deflagration to detonation. Imagine a finely grained powder which is ignited on the surface. As the surface burns the deflagration flame will travel through the space between grains to some inner grain. It is feasible that some of the inner powder grains might ignite producing their own gaseous products and liberating heat. The action of the inner burning tends to increase the pressure in that area since the gas formed can not escape as quickly as it is formed due to the small channels available for its flow between grains. Hence the pressure builds up, and with it the rate of burning. The action can be likened to the movement of a piston on a cylinder of gas. Eventually this build-up forms a shock wave which travels with sonic velocity. There is a temperature rise greater than isentropic along with the shock and this wave acts similar to an impact load hitting an explosive. It is known experimentally that an explosive can be detonated by impact. The evidence points to three influences:

1. Adiabatic compression of minute gas bubbles
2. Friction between explosive particles
3. Heating by viscous flow at impacting surface

These all tend to raise the temperature; there occurs deflagration which turns into detonation.

It is proposed that the temperature rise and shock wave traveling with the speed of sound causes the consumption of the entire mass instantly; and it is this sonic velocity of the wave that gives the discontinuity between deflagration and detonation.

As confirmation of the theory there exists a critical mass of granular explosive above which deflagration may become detonation; below which there is a rising rate of deflagration followed by a falling rate of deflagration. This may be of the order of one ton for TNT, and as small as milligrams for other chemical compositions. When an explosive which is easily detonated is pressed together under high pressure, thereby reducing the space between grains, a deflagration may result. Pressures of 5000 psi are needed for mercury fulminate; 50,000 psi for lead azide.

Kistiakowsky shows that confinement alone is not sufficient to cause detonation; there are other factors such as geometric shape. Confinement and the subsequent pressure rise increase the burning rate. This is explained by considering two reactions occurring as the powder burns. A primary reaction occurs forming gases which then react liberating great amounts of heat. This heat from the second reaction must be transferred to the surface of the powder to continue combustion. At high pressures (under confinement) the molecules are traveling faster or there are more of them per unit volume and the heat may be transferred more readily.

It is concluded that with powder propellants a maximum pressure exists for a particular mass of powder above which detonation will probably occur. Very high pressures have been attained in rockets but not for burning times of the order of 10 seconds. Since the time of firing is at least equally as important to this discussion as the pressure reached (for a wind tunnel would be worthless were there insufficient time to obtain readings), evidence supports the conclusion that 6000 psi is the maximum pressure practicable.

Initial experiments with liquid propellants led to many detonations on test runs. It is known that detonations exist in internal combustion engines with liquid fuels and that a fuel being tested as a possible rocket propellant is given many types of impact tests to test its sympathetic detonation and the possibility of this detonation wave carrying through small fuel lines. (44) Since the running time is of the order of minutes, other factors are possible. One is the vibrations occurring during burning. (45) Another is the possibility of burn-out of the metal wall due to improper cooling or temperature distribution in the wall. (46)

This last problem, that of the heat transfer to the wall, has been the main purpose of the investigations of Zucrow and Beighley (47) at Purdue. They have built model rockets burning WFNA-JP-3 fuel and operated them at various pressures, and intended to test to 2000 psi. Results published in 1952 confirm the suspicion that heat transfer in the nozzle will be critical at high pressures. First experiments were conducted with stainless steel nozzles but later copper was used since the steel eroded too much. The average difference in throat area before and after each run with copper was approximately 0.4 per cent.

Liquids present other problems. Many of them are difficult to handle and leaks may become dangerous. The higher the chamber pressure, the higher must be the pressure of the fuel injection system.

Use as a Wind Tunnel

Years ago before hypersonic wind tunnels were discussed, the National Advisory Committee for Aeronautics tested a model airplane in one of their subsonic tunnels; then took an actual plane which practically duplicated the model with the same type of instrumentation and flew it to determine the actual correlation between model tests and actual flight conditions. (48)

Since hypersonic work is still in its infancy such correlation check has not been made, but an attempt has been made to evaluate the different flow parameters and their importance for similarity of flow conditions. As experimental results are evaluated more will be known as to the effects of the various parameters.

In general, the following is considered an all-inclusive list for similarity:

1. Geometrically similar including scale reproduction of all small details such as surface roughness (important in boundary layer investigations and force distributions).
2. Reynolds number
3. Froude number
4. Mach number
5. Prandtl number
6. Grashof number
7. k, the ratio of specific heats

Usually in wind tunnel work Froude number is eliminated from discussion since no free surface is involved. It becomes important in ship hull design. Grashof number is not considered since gravity effects are negligible. (49)

If the test fluid is air, the variations of k and Prandtl number are sometimes considered unimportant. Such is not the case at hypersonic speeds. Chapman (50) has investigated the use of fluids other than air in wind tunnels and has shown (Figure 13) that for hypersonic work the variation with temperature of the k for air under flight conditions may be appreciable while k for air in a wind tunnel is quite constant. Instead of showing the computed variation as a function of a temperature parameter, an enthalpy parameter $(H - H_\infty)/(H_t - H_\infty)$ is employed which always is zero for free-stream conditions (subscript ∞) and always is unity for reservoir conditions (subscript t). It is noted that gases with k significantly different from air would not yield in a hypersonic flow at a given Mach and Reynolds number, either the same skin friction or the same pressure distribution as air would yield. Prandtl number P_r is a dimensionless heat transfer parameter. Heat transfer from the high temperatire skin of a plane flying at supersonic speeds can alter the boundary layer surrounding that skin. Boundary layer variations can change shock waves and Mach numbers.

Discussion of the effects of differing values of k and Pr seem ultimately to lead to a consideration of heat capacity lag effects. Kantrowitz (51) has discussed some aspects of this problem. He calculates that the drag coefficient of a wing in pure carbon dioxide might be twice as large as the same wing tested in air with the same Mach and Reynolds numbers. It all depends on the relaxation time of the gas in question.

Gunn (52) explains that the specific internal energy of a gas may be denoted by $E = E_t + E_r + E_i$ ignoring the energy content of electronic excitation and molecular dissociation,

where E = specific internal energy of the gas

E_t = kinetic energy of translation of molecules

E_r = the energy of molecular rotation

E_i = the energy of vibration of molecules

The translational and rotational energies of a gas very quickly adjust to their equilibrium values (in the space of a few molecular collisions). They are assumed to have their equilibrium value and are called active degrees of freedom. The vibrational degrees of freedom are relatively much more slowly adjusted, oxygen taking over 500,000 collisions at room temperatures. For this reason vibrational degrees of freedom are classified as inert. The time necessary for the molecule to adjust itself, the relaxation time, varies with different gases. Maximum times of the order of 10^{-4} seconds are mentioned.

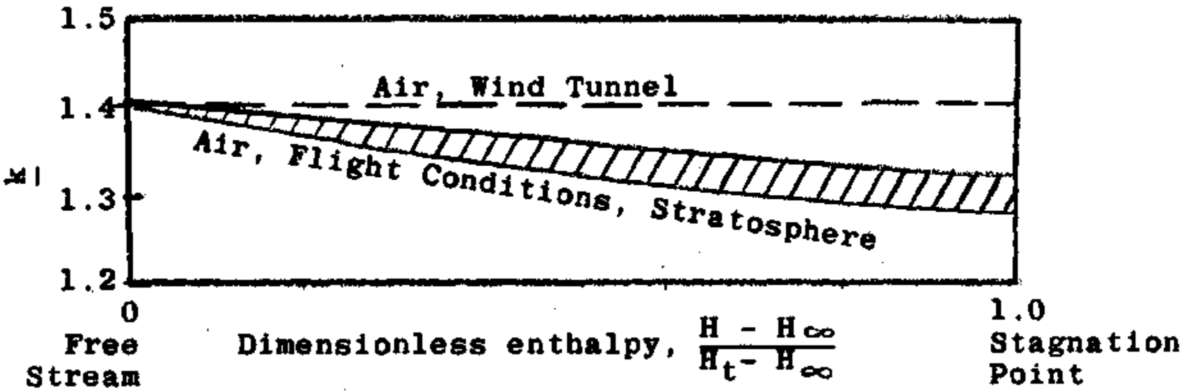

Figure 13. Variation of Specific-heat Ratio Between Free-Stream and Stagnation Conditions at Mach Number of 5.

Gunn states that a shock wave is no longer a discontinuous transition, but consists of a discontinuous change involving only the active degrees of freedom followed by a continuous change extending over a time of the order of the relaxation time; i.e., over a distance of the order of the vibrational mean free path.

Stated another way, consider a gas expanding through a channel such as a wind tunnel with velocity increase and temperature decrease. As the gas flows, while the energy in the "active" degrees of freedom immediately adjusts itself to the rise in velocity and fall in temperature, the energy in the vibrational or "inert" modes lags behind its equilibrium value -- in fact, continually remains above its equilibrium value, as the temperature falls. This effect is most pronounced when the change of temperature takes place in times of the same order as the relaxation time of the gas. Thus, during the process of flow, the effective temperature T_i of the "inert" modes will be constantly above the temperature T of the "active" modes, and there will, as a consequence, be an irreversible flow of heat from the "inert" to the "active" modes with a resultant increase in the entropy of the gas as a whole. This increase in entropy is accompanied by a diminution of available energy, and as a consequence lower absolute velocities at any section of the wind tunnel would be expected than if the relaxation time were zero. When an obstacle is placed in the flow the entropy rise has associated with it a drag.

Gunn, in discussing the possibility of using sulphur hexaflouride as a wind tunnel fluid rather than air, compares the effect of different relaxation times of the two fluids. He calculates conditions immediately behind a shock and compares them with conditions after the vibrational mode has come to equilibrium. This distance he terms shock wave thickness and estimates its magnitude of the order of millimeters. Taking Mach number of about 2 and the temperature in each case the same, variations of temperature, pressure, and density immediately behind the shock as compared with conditions after the shock wave defined above are negligible for air. With sulphur hexaflouride, temperature varies about 75 per cent, density about 55 per cent, and pressure about 10 per cent.

Another consideration in the use of the rocket exhaust as a wind tunnel is the strength of the model placed in such flow. From a mechanical strength standpoint, the aerodynamic force acting on the front of the model will be proportional to $\frac{\rho V^2}{2} \times A$ where ρ and V are density and velocity of the fluid, A is the frontal cross-sectional area of the model. It has been discussed from different viewpoints by Hilton (53) and Pankhurst (54). One is concerned with the deformation of the model and the difference between the model deformation and the plane deformation in flight. Differences here will change the aerodynamic force pattern for instance over a wing surface and similarity will no longer hold. Another author concludes that at high Mach numbers the interference effect of the model support will be significant, and that a tunnel of high density will demand greater supports. It can be seen that the force acting on the front of the model is increased by increasing the density of the fluid, all other things being the same.

The proposed wind tunnel encourages the use of high stagnation pressures. The fluid density in the test section is a function of the fluid density in the rocket chamber. This density increases with the pressure. Therefore, it is concluded that there is a limiting pressure in the rocket chamber above which a model placed in the test section would fail.

The suggestion seems in order to make the model smaller, thus reducing the frontal area and increasing its ability to withstand deformation. This presents another problem. From similarity considerations for compressible flows the Mach number must be reproduced. It is desirable to test at full scale Reynolds number, Re. In subsonic work failure to test at flight Re resulted, in at least two cases, in errors. The drag of airships was shown to be distinctly a function of shape at model Re but much less critical at full flight, Re. Again if absolute values of maximum lift and minimum drag are wanted, then Re must be equal to full scale. (55) Since Re is proportional to l, a characteristic length of the model, any reduction in size of the model reduces the Re proportionally.

The thermal strength of the model must be considered. Kaye (56) has calculated that a missile flying at 50,000 feet altitude at Mach number of 8 will have a temperature of $4000^{o}F$ at its nose. Wind tunnels using atmospheric air as the test fluid seldom have stagnation temperatures higher than $1000^{o}F$. Calculated temperatures in rockets are of the order of $5000^{o}F$. As this high-temperature fluid flows through the nozzle the thermal energy will reduce while the velocity increases. At the nose of the model where the velocity again is reduced to zero, temperatures should be of the order of $5000^{o}F$. This appears to the author to be possibly a blessing in disguise. Tests made in conventional tunnels at the low stagnation temperatures give temperatures in the test section well below room conditions. As a result, to eliminate heat transfer effects models are sometimes cooled before test runs so they will be at the temperature of the test section with a uniform temperature distribution. However, in flight the temperature distribution is not uniform. Also in flight there is radiant heat transfer from the missile surface to the surroundings. At the high stagnation temperatures of the proposed tunnel, the walls of the test section might be refrigerated to simulate this radiation, and to give temperature distributions similar to those of flight.

It is concluded that while model strength is a consideration in such a tunnel design, it is not hopeless. Tunnels today are built with stagnation pressures of 3500 psi, and 5000 psi is discussed. Should tests have to be run at Re less than full scale, it still is unknown what the effect will be; perhaps it will be negligible; perhaps the effect will only be felt by certain of the results obtained.

CHAPTER VI

COMPARISON OF COST AND CONSTRUCTION

It would be unfair to compare this proposed tunnel with a continuous-operating hypersonic tunnel such as the one built at Cal Tech by the Navy and Ordnance Departments.

The National Advisory Committee for Aeronautics has built at Langley Field an intermittent hypersonic tunnel capable of Mach of 7. The test section is 10in x 11in. Air is stored at 50 atmos in 400 cu ft tank. The capacity of the vacuum tank is 12,000 cu ft. A heater heats the air to a max of 1300 R, and a cooler after the test section increases the effectiveness of the vacuum tank.

At M.I.T. there is a tunnel designed for Mach of 7.2. Air is dried and compressed to 3000 psi by a four-stage compressor. It is stored after being dried, in 11 bottles, each of 34 cu ft capacity. Temperatures up to about 1300 R may be obtained with an aluminum wire heater (1000 lbs of aluminum wire). A steam ejector system is used on the exhaust side to reduce the pressure to 1.5 psia.

Perhaps the best comparison may be drawn with North American Aviation's tunnel built in Los Angeles. (57) This tunnel is designed to provide a Mach number of 5.3 in a square test section 16 inches on a side. It is intermittent in operation. Dry air is stored at high pressure in a 22,500 cubic foot tank; a 36,000 cubic foot vacuum tank is exhausted to a 99.8 per cent complete vacuum. Its time of operation is from 15 to 20 seconds. (58) The tunnel is estimated to represent an investment of $286,000. The power cost to operate such a tunnel would be a function of the time available between runs to build the high pressure and high vacuum.

An idea of rocket cost may be gotten from the cost of mass produced rockets used by the Army. The 3.5 inch rocket weighing 17 pounds is estimated at $13.60. The 4.5 inch rocket costs $45. The weight of charge in the 4.5 inch rocket is about 5 pounds. (59) The weight of a typical charge required for the proposed tunnel is calculated in the appendix of this paper as 71 pounds of powder. Liquid rockets with the pumping facilities required are higher. An average figure for liquid fuel cost is thirty cents per pound.

A true cost comparison should include such nebulous items as cost of the land required and how long the tunnel may be used before replacement is necessary. At first glance the physical size of the rocket-powered tunnel would be appreciably smaller than conventional. However, for safety of operation it may be necessary to place the rocket tunnel away from buildings and the land investment could possibly be similar in each case.

Item for item the rocket powered tunnel will be less expensive. Comparing Figures 5 and 7 and recalling that the diffuser will be eliminated from the proposed tunnel, the following items will be surplus: compressor, cooler, settling tank,dryer, compressed air reservoir, heater, vacuum tank, and vacuum pump. The initial cost of the compressor, tanks, and vacuum pump will be dependent on their capacity. In order to have as little delay as possible between runs this cost is high, as is the power cost per run. If many runs are required, this delay time may become an important factor in tunnel design. With the replacement of a rocket or recharging of fuel tanks as the only requirement between runs, it is concluded that the proposed tunnel would save both time and money.

CHAPTER VII

DISCUSSION

From nozzle theory, given high enough pressure and temperature ratios, the flow of Mach number of five may be realized. From rocket theory it is feasible to burn fuel fast enough so that any pressure may be obtained. It therefore seems that a tunnel as proposed could exhaust to the atmosphere and develop hypersonic flow.

Actually there are several limiting factors. The value of k for the exhaust gases of rockets is about 1.2. This would require a chamber pressure of 27,000 psia. Experience with rockets sets 6000 psia as the upper limit for this pressure without detonation. This demands the exhaust of the tunnel be done at a pressure of 0.25 atmospheres or less.

The addition of a diffuser on the end of the test section could lower the pressure ratio required to maintain the flow once started. The possibility of using a high-pressure rocket of short duration to drive the initial shock and then a ten second lower-pressure rocket to maintain the flow seems attractive. However, the maximum gain is seen to be thirteen per cent at Mach number of five and then the diffuser would not be of much benefit since to realize this gain it is necessary to neck down the second throat to a smaller area after the shock has passed the test section. With a ten second flow there would be insufficient time to make such adjustments.

Since the gain in the diffuser is so low it appears that use of a second rocket or bank of rockets as an ejector on the exhaust could act to hold the pressure at 0.25 atmospheres for the ten second run. This would require two or more ten-second rockets to be burned for one test run. Operating costs would increase.

While actual estimates of cost are difficult to compute, from a consideration of the factors involved and the size of the equipment to be built and maintained it appears that a rocket-powered tunnel would be much less expensive than those currently in use.

Such a tunnel could feasibly be used for fundamental research of compressible fluid flow at hypersonic speeds. However, the use of this tunnel to test plane or missile designs has its advantages and disadvantages. The major disadvantage is that a test run in one fluid is not always similar to flight in another. But for that matter, tests run in wind tunnels using air are not similar at hypersonic speeds in many respects to flight in air at high altitudes. In some ways the similarity with the rocket tunnel will be improved over conventional tunnels. An example of tunnel tests at subsonic speeds in which the fluid was not air is given by Great Britain. Shortly after the last war a study was made of the possibility of using a jet engine as inductive power to draw air through a wind tunnel. (60) Such a tunnel was later built. In order to prevent condensation of the moisture in the air some of the exhaust of the engine was mixed with the incoming tunnel air to raise its temperature. (61)

Certain questions have been considered in this study and they are believed to be among the most critical. Others remain. There is a question of volume of the rocket necessary to ensure completeness of combustion in the rocket motor. There is the question of smoke in the exhaust. In 1948 Kimball (62) of Aerojet Corporation stated that a smokeless JATO motor was just on the horizon. If it has been developed, is the exhaust clear enough to take Schlieren pictures?

The question of using rockets as ejectors remains important. Scientists at the University of California (63) have proposed the use of six rockets as ejectors to draw air through a test section to obtain a relatively cheap wind tunnel capable of simulating flight at 400 mph, altitude of 5000 to 60,000 feet. Six rockets, each of 5000 pound thrust, were to be mounted concentrically around the area ahead of the diffuser. General Electric was to work on the design of such rockets. Perhaps such ejector use coupled with the rocket exhaust as test medium will be the solution to some hypersonic test problems.

CHAPTER VIII

CONCLUSIONS

1. A rocket exhausting through a converging-diverging nozzle can theoretically attain hypersonic flow.

2. The exhaust gases of rockets have a specific-heat ratio k, of the order of 1.2.

3. The pressures attainable in restricted powder-burning rocket motors are limited to about 6000 psia as a maximum before detonation occurs.

4. With this pressure maximum and k of 1.2 a pressure of 0.25 atmospheres would have to be maintained in the exhuast to achieve a flow of Mach number five (based on perfect gas in one-dimensional isentropic flow).

5. A diffuser added to such a tunnel operating for ten seconds would be of little assistance.

6. Such a tunnel with its high stagnation temperatures would have no difficulties of moisture or gas condensation.

7. Such a tunnel could be used for fundamental research but its usefulness in missile or plane design is not certain.

CHAPTER IX

RECOMMENDATIONS

It is recommended:

1. That the use of rockets as ejectors be further investigated with application to this proposal.

2. That the effect of erosion in the nozzle on wind tunnel tests be further investigated.

APPENDIX

Sample Calculations

Assume: P_c = 6000 psia

a = 0.28 in/sec b = 0.00037 in^3/lb-sec	These are experimental data for normal fast powders; specifically for powder ROW 10015 @ 27°C, reference (20), pg. 47.
ρ = 0.059 lb mass/in^3	Typical for most powders.
C_d = 0.007	Typical for rockets using smokeless powders; (20), p. 42.

Test section is circular, diameter of five inches.

Test section Mach number equals five.

Duration of run: 10 seconds.

Cigarette burning desired.

k = 1.2 for exhaust gases.

1. Velocity of travel of burning surface:

$$r = a + bP_c$$
$$= 0.28 + 0.00037\ (6000)$$
$$= 2.52 \text{ in/sec}$$

2. Throat area required:

For isentropic flow of perfect gas, ref (3), p. 86,

$$\frac{A}{A_t} = \frac{1}{M}\left[\left(\frac{2}{k+1}\right)\left(1 + \frac{k-1}{2} M^2\right)\right]^{\frac{k+1}{2(k-1)}}$$

$$= \frac{1}{5}\left[\left(\frac{2}{1.2+1}\right)\left(1+\frac{1.2-1}{2}(5)^2\right)\right]^{\frac{1.2+1}{2(1.2-1)}}$$

$$= 116$$

$$A_t = \frac{A}{116} = \frac{19.6}{116} = 0.169 \text{ in}^2$$ or a throat of circular section, diameter of 0.464 in.

3. Discharge rate of gases produced:

$$\frac{d\ M_d}{dt} = C_d A_t P_c = 0.007\ (0.169)(6000)$$
$$= 7.1 \text{ lbs/sec.}$$

4. Burning surface required:

$$\frac{d\ M_b}{dt} = \frac{d\ M_d}{dt} = 7.1 \text{ lbs/sec} = S\rho r$$
$$= S\ (0.059)(2.52)$$
$$S = 47.8 \text{ in}^2\text{; radius} = 3.9 \text{ in.}$$

5. Powder mass required:

$$7.1\ \frac{\text{lb.}}{\text{sec}} \times 10 \text{ sec} = 71 \text{ lb. powder}$$

6. Length of powder charge (cigarette burning):

$$2.52\ \frac{\text{in}}{\text{sec}} \times 10 \text{ sec} = 25.2 \text{ inches}$$

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